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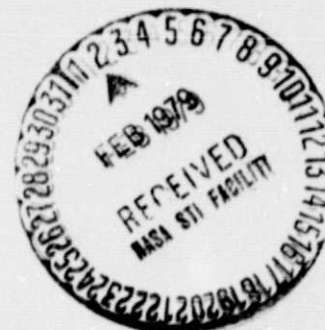
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NASA THERMAL BARRIER COATINGS -
SUMMARY AND UPDATE

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NASA THERMAL BARRIER COATINGS - SUMMARY AND UPDATE

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ABSTRACT

The work conducted at the NASA Lewis Research Center to evolve and evaluate a thermal-barrier coating system will be discussed. A durable, two-layer plasma-sprayed coating consisting of a ceramic layer over a metallic layer was developed that has the potential of insulating hot engine parts and thereby reducing metal temperatures and coolant flow requirements and/or permitting use of less costly and complex cooling configurations and materials. The paper summarizes the results of analytical and experimental investigations of the coatings on flat metal specimens, turbine vanes and blades, and combustor liners. Discussed are results of investigations to determine coating adherence and durability, coating thermal, strength and fatigue properties, and chemical reactions of the coating with oxides and sulfates. Also presented are the effect of the coating on aerodynamic performance of a turbine vane, measured vane and combustor liner temperatures with and without the coating, and predicted turbine metal temperatures and coolant flow reductions potentially possible with the coating. Included also are summaries of some current research related to the coating and potential applications for the coating.

INTRODUCTION

It has been recognized for a long time that ceramic coatings could insulate engine parts from hot combustion gases. About 25 years ago, coatings were tried (refs. 1 and 2) as a means of reducing metal temperatures of, and providing corrosion protection for, turbine blades. These early tests, however, were made at relatively low turbine inlet temperatures and heat fluxes, and the use of the coatings showed no significant benefits. However, the higher gas temperatures and pressures of current and future gas turbine engines provide environments where benefits may be realized by the use of ceramic thermal-barrier coatings. These benefits include reducing component cooling require-

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ments and metal temperatures and thereby improving engine performance and durability. These considerations plus recent improvements in coating deposition techniques have stimulated a renewed interest in the coating of hot parts. A two-layer thermal-barrier coating (TBC), consisting of a stabilized zirconia coating over a metal bond coating, has been developed at the NASA Lewis Research Center and is under investigation.

This paper summarizes the results of analytical and experimental investigations of thermal-barrier coatings on flat metal specimens, turbine vanes and blades, and combustor liners. The paper discusses the results of investigations to determine: coating adherence and durability, coating thermal, strength and fatigue properties, and chemical reactions of the coating with oxides and sulfates. Also presented are the effect of the coating on the aerodynamic performance of a turbine vane, measured vane and combustor metal temperatures with and without the coating, and predicted turbine metal temperatures and coolant flow reductions potentially possible with the coating.

ANALYTICAL AND EXPERIMENTAL RESULTS

Coated Specimens and Turbine Blades

An air-cooled turbine blade covered with a ceramic thermal-barrier coating is shown in figure 1. The metal substrate was prepared by grit blasting. A layer of the bond coating was then deposited by arc plasma spraying, followed by a layer of ceramic which was also arc plasma sprayed. Details of the coating procedure and composition are given in references 3 to 5. The thickness of the bond coating was nominally about 0.010 cm and that of the ceramic between 0.025 and 0.064 cm. The majority of the ceramic coatings were zirconia stabilized with yttria. Initial tests included zirconia stabilized with calcia and magnesia. The bond coating material was primarily NiCrAlY.

Initial tests of the coating adherence on a variety of nickel and cobalt-base alloys were conducted in a commercial electric furnace. The results of these tests (ref. 3) gave encouragement on the adherence capability of thermal barrier coatings. The tests helped establish the surface cleaning and coating procedure and disclosed that the ceramic-bond interface has a limiting temperature of about 1367 K for good adherence.

Analyses were made and tests conducted in an existing research engine (ref. 3) to determine the benefits attainable with coated turbine vanes and blades. A simple one-dimensional, steady-state heat balance through a composite wall consisting of the ceramic, the bond coating, and the metal wall was

used. The results of the analysis and tests on coated vanes in an existing re-search engine are shown in figure 2. The results show good agreement (within 3.5 percent) between prediction and measurement and show that large reductions in metal temperatures can be obtained with the coating. At a coolant-to-gas flow ratio of 0.06, for example, the metal temperature was reduced by 190 K, from 1055 K for the uncoated vane to 865 K for the coated vane.

Greater benefits of the thermal-barrier coating occur at the high heat fluxes of cooled hardware that exist in the high-gas-temperature- and pressure- environments of core turbines of advanced turbofan engines (ref. 3). In this environment, bulk turbine-vane metal temperatures (integrated average temperature over the entire vane) were predicted to be reduced by as much as 390 K with a 0.051-cm thickness of zirconia. Another calculation comparing a full-coverage film-cooled turbine and a thermal-barrier-coated convection-cooled turbine (ref. 6) showed that the thermal-barrier-coated turbine has the potential for operating at coolant flows similar to those of a full-coverage film-cooled turbine.

Aerodynamic performance tests. - Tests were conducted (ref. 7) in a two-dimensional cold-air cascade to determine the effect of the coating on the aerodynamic performance of turbine vanes. The results of these tests, shown in figure 3, indicate that the kinetic energy loss coefficient for the as-sprayed-on coated vane was about 4 points or $2\frac{1}{2}$ times larger than that for the uncoated vane at a design exit velocity ratio of about 0.8. Smoothing the coating with aluminum oxide reduced its surface roughness (from 8.9 to $2.3\ \mu\text{m}$) and eliminated most of the additional loss. The smoothed coated vane had a kinetic energy loss coefficient that was still 0.7 of a point larger than that for the uncoated vane. This loss was attributed to the 38-percent-thicker trailing edge caused by the coating.

Furnace tests. - Long-duration cyclic durability tests were run with coated, flat metal specimens in a furnace. In these tests (ref. 5) the coated specimens were heated in air to 1248 K for 1 hour (specimens reached temperature in about 4 min) followed by furnace cooling to 553 K within 1 hour. The results of these tests summarized in table I show that $\text{ZrO}_2\text{-}12\text{Y}_2\text{O}_3$ (weight percent) was the most adherent, followed by $\text{ZrO}_2\text{-}3.4\text{MgO}$, partially stabilized $\text{ZrO}_2\text{-}5.4\text{CaO}$, and zirconia totally stabilized with CaO. It is evident from the data in this table that the properties of the substrates (such as coefficient of thermal expansion) apparently had little effect on the adherence or performance of the coatings.

Engine tests. - A research turbojet engine at the NASA Lewis Research Center was used to evaluate the durability of the coatings in a gas turbine environment.

Steady-state durability tests (ref. 3) were conducted on 0.028-cm-thick calcia-stabilized zirconia coating on two vanes and two blades as part of another

research test. The coated vanes and blades were usually operated at turbine inlet gas temperatures of 1367 to 1644 K and a gas pressure of 0.3 MPa. The resulting coated vane and blade leading-edge metal temperatures generally did not exceed 920 K. On several occasions, hot starts resulted in transient metal temperatures of 1200 K. At the completion of 150 hours of test time, including 35 start-and-stop cycles, the coatings showed no evidence of deterioration.

Cyclic durability tests of three zirconia coatings stabilized by either calcia, magnesia, or yttria were conducted (ref. 8) on 74 air-cooled turbine blades in the research engine. The ceramic coatings were about 0.038 cm thick, and the NiCrAlY bond coatings were about 0.010 cm thick. The gas temperature was held at 1644 K (full power) for about 70 seconds, and then decreased in 20 seconds to 1000 K (flameout). The combustor was then reignited, and the engine reached full-power conditions in 30 seconds. A total of 500 of these 2-minute cycles were run. The coatings (except for minor foreign object damage incurred during the first 100 cycles) showed no visual evidence of deterioration. A trailing-edge view of the rotor assembly of the coated blades after conclusion of the tests is shown in figure 4. The two uncoated blades shown in the figure were used as reference blades for other tests. The black spots on the blade tips are soot deposits that accumulated during engine shutdown. The black lines along the span near the root on the suction surface were also caused by soot deposition.

The microstructures of the bond coating and the ceramic coatings were metallographically examined on several blades at the leading-edge region, where durability problems are most likely to occur. The ceramic microstructure as seen in figure 5 consisted of solid material connected with a network of fine voids interspersed with larger voids. The photomicrographs also show that the aluminide that was originally present on all blades was not completely removed by grit blasting.

Hot-gas-rig tests. - In these tests (ref. 5) hot gas at a Mach number of 0.3 impinged on the leading edge of a coated test blade held in a fixture as shown in figure 6. Internal flow of air was used to cool the blade wall to desired metal temperature levels. Jet A fuel was used for this rig. Another test (ref. 9) used a burner rig that was fired with natural gas that impinged on coated specimens at a Mach number of 1. Two cycle times were investigated with the Mach 0.3 rig. In the short cycle the hold time in the hot-gas stream was 80 seconds, and in the long cycle the hold time was 1 hour. The time for heat-up or cool-off in both these cycles was about 30 seconds. The cyclic tests in the Mach 1 rig had the 1-hour hold time.

The results of the short cycle tests in figure 7 show that the yttria-stabilized zirconia was superior to the other two coatings. The 0.028-cm thick yttria-stabilized zirconia satisfactorily completed 2000 cycles at a surface temperature of 1463 K when testing was stopped. The magnesia-stabilized coating of the same thickness completed 1010 cycles when testing was stopped due to excessive erosion. The results with the 0.038-cm-thick coatings at a coating surface temperature of 1553 K show that the yttria-stabilized coating completed 3200 cycles before testing was stopped because of excessive erosion. The tests of the calcia-stabilized zirconia were stopped after only 200 cycles because of failure near the ceramic-bond material interface.

The results of the long-cycle tests of 0.051-cm-thick yttria-stabilized zirconia coatings are shown in figure 8. These results show that the coating had completed 246 one-hour cycles at a substrate (metal) temperature of 1173 K and a coating surface temperature of 1683 K before the test was stopped because of excessive erosion. This erosion is attributed to carbon from the Jet A-fueled burner. In the tests carried out in the natural gas fired Mach 1 burner rig (ref. 9), erosion was not evident.

Figure 8 shows that coating life is significantly reduced with increased coating surface or substrate temperatures. Increasing the coating surface temperature by 130 K (substrate temperature by 60 K) resulted in a fivefold decrease in cyclic life (from about 250 to 50).

Research has recently been conducted at NASA (ref. 10) on the effect of changes in yttria content in the bond and ceramic coating layers on adherence and durability. Results (ref. 10) indicated that the best durability is obtained with yttria content by weight of 6 to 8 percent for the ceramic and yttrium content of 0.15 to 0.35 percent for the bond coating. As an example, coated air-cooled blades tested in the natural gas-fueled burner rig showed that the TBC with 0.35 percent yttrium in the bond coat and 7.9 percent yttria in the ceramic withstood 1500 one-hour cycles at a surface temperature 1823 K without failure while the TBC with 0.61 percent yttrium in the bond coat and 11.5 percent yttria in the ceramic failed after 1120 cycles at a temperature of 1663 K. The majority of the tests described herein had a NiCrAlY bond coat with nominally 0.6 percent by weight of yttrium and a zirconia ceramic with nominally 12 percent by weight of yttria.

High cycle fatigue tests. - Tests were run to determine the effect of a thermal-barrier coating on the high cycle fatigue life of turbine blades. These tests were run as a part of a study to determine the advisability of installing several coated turbine blades in an advanced high-gas-temperature- and pressure-turbofan test engine. One blade with a 0.036-cm-thick yttria-stabilized zirconia thermal-barrier coating and three uncoated blades were

subjected to high cycle stress loading while being heated in an electric furnace to 1033 K. The results of these tests, wherein the blades were run until failure, showed that the coated blade had about 10 times greater fatigue life than the uncoated blades. The failure origin of the coated blade was in the blade metal wall material, and the coating did not appear to assist in the initiation or propagation of the failure. The reason for the substantial improvement in the fatigue life of the coated blade as compared with the uncoated blades has not been determined.

Low cycle fatigue tests. - In these tests (ref. 11), self-resistance-heated specimens were strain cycled by applying displacements at the ends. The specimens were made of TAZ-8A and the testing was conducted in air at specimen temperature of 1193 K. The coatings applied to the specimens were (1) yttria stabilized ZrO_2 and NiCrAlY, (2) the latter with an overlay of Al_2O_3 , or (3) NiCrAlY alone. The results of the tests at 1193 K indicated that ZrO_2 coated specimens had about four times the fatigue lives of the uncoated or NiCrAlY alone coated specimens. Analysis further indicated that the resistance of the ZrO_2 to low cycle fatigue is not expected to deteriorate even to 1590 K. The addition of the outer coat of Al_2O_3 to the TBC did not influence fatigue resistance.

Adhesive/cohesive strength tests. - Tests (ref. 12) conducted at both elevated and room temperature indicated that the weakest link in the TBC was the oxide/NiCrAlY interface region adhesive strength, and the cohesive strength of the first deposited ceramic layers. At room temperature, the interface region had a strength of 6.2 MN/m^2 which was about one-fourth of that of the layer materials. The zirconia ceramic had a cohesive strength of 24.6 MN/m^2 and the NiCrAlY bond coating had a cohesive strength of 25.1 MN/m^2 . The NiCrAlY and ceramic failed primarily at interparticle boundaries. Tensile and compression tests conducted at 1223 K showed the TBC failed in a manner similar to the room temperature tests.

The low adhesive strength at the oxide/NiCrAlY interface and the low cohesive strength of the first deposited oxide layers is attributed (ref. 12) to the deposition procedure. The reasoning was that the first oxide layers when plasma sprayed on a cold high-thermal-conductivity substrate suffers greater thermal shock than successive layers deposited on the warm low conductivity oxide (steady state temperature of about 523 K). This suggests that deposition of the oxide on a preheated substrate may strengthen the oxide/NiCrAlY interface region of the thermal barrier coating.

Reactions with oxides and sulfates. - Various chemical compounds representing impurities in fuels and combustion gases as well as elements of the bonding alloy were reacted (ref. 13) with zirconia with 8 percent by weight of yttria.

The reactions were studied at 1473, 1573, and 1673 K and times of 800, 400, and 200 hours, respectively. The results showed that Na_2SO_4 , K_2SO_4 , Cr_2O_3 , Al_2O_3 , and NiO did not react with the zirconia, but that reactions were obtained with CoO , BaO , BaSO_4 . The compounds that reacted preferentially for the predominant cubic phase of the zirconia were V_2O_5 and P_2O_5 and that for the monoclinic zirconia phase were N_2O , K_2O , CoO , Fe_2O_3 , MgO , SiO_2 , and ZnO . The results of these tests indicate for example that it might not be advisable to use CoCrAlY or FeCrAlY type materials as bond coating materials, or use yttria stabilized zirconia in an iron oxide environment, or with fuel additives containing barium.

Thermal conductivity measurements. - Measurements were made of the thermal conductivity of arc-plasma sprayed layers of NiCrAlY and yttria stabilized zirconia in references 14 and 15. The thermal conductivity of the zirconia, k_z , over a temperature, T , range of 400 to 1900 K (as determined from these data in ref. 16) varied from about 0.50 to 1.4 W/mK and is represented by the following equation:

$$k_z = -1.597 + 0.011 T - 1.896 \times 10^{-5} T^2 + 1.324 \times 10^{-8} T^3 - 3.077 \times 10^{-12} T^4 \quad (1)$$

The thermal conductivity of the NiCrAlY bond coating, k_n , varied linearly from about 5 to 16.0 W/mK between temperatures of 367 to 1367 K and is represented by the equation

$$k_n = 9.41 \times 10^{-3} T + 2.89 \quad (2)$$

Thermal emittance and absorptance measurements. - Normal spectral emittance measurements were made on the TBC in reference 17. The normal spectral emittance of the TBC was obtained at a single NiCrAlY coating thickness of 0.010 cm and at ceramic coating thicknesses of zero to 0.076 cm at wavelengths of 0.4 to 14.6 μm and at temperatures of 300 to 1590 K. Figure 9 shows that the emittance of the coating varies greatly with wavelength and ceramic thickness. This change in emittance with thickness is attributed to the translucence of thin layers of plasma-sprayed ceramic in the lower wavelength range of about 1 to 5 μm . However, the emittance did not change with zirconia stabilizer, roughness, color or view angle. Much of the incident gas radiation in a turbojet combustor is in the lower wavelength range (1 to 3 μm) and therefore the coating will reflect most of this radiation. Equations were derived for calculating the total hemispherical absorptance and emittance of the TBC at temperatures of 700 to 2800 K and ceramic thicknesses of zero to 0.0762 cm. These equations and the corresponding data are shown in figure 10. These correlating equations are in

a form convenient for use by designers in the calculation of radiative heat loads.

Coated Combustor Liners

Tests were conducted on the thermal performance of the TBC on a single-can combustor from the JT8D engine over a range of gas temperatures and pressures that included engine idle, cruise, and takeoff (ref. 18). Tests were run with two fuels of widely different aromatic content. One fuel was ASTM A-1 (Jet A) and the other was a blend of Jet A and a mixture of single-ring aromatic compounds. The thermal-barrier-coated combustor liner is shown in figure 11.

The effect of the coating was to significantly reduce metal liner temperatures. Maximum liner temperatures as a function of average exhaust-gas temperature for Jet A fuel are shown in figure 12. At an exhaust-gas temperature of 1325 K, representative of takeoff conditions, the maximum liner temperature was reduced from about 1220 K for the uncoated liner to about 1060 K for the ceramic-coated liner. At an exhaust-gas temperature of 1126 K, representative of cruise, the maximum liner temperature was reduced from about 1050 K to about 920 K through the use of the ceramic coating. The maximum liner temperatures were higher with the high-aromatic-content Jet A fuel, but the liner metal temperature reductions with the coating was approximately the same with both fuels. The maximum uncoated liner temperature at takeoff with the blended Jet A was about 1265 K as compared with 1050 K with the coated liner. Results of tests on an annular combustor (ref. 19) also showed large reductions in metal temperatures by the use of the TBC.

The results of measurements presented in reference 18 also showed a substantial reduction in flame radiation due to the higher reflectance of the ceramic coating as compared with the bare metal wall. Figure 13 shows that with Jet A fuel at takeoff condition (1325 K), flame radiation was reduced from 6.9 to 6.1 W/(cm²/sr). Apparently, the radiation reflected from the coated liner walls back into the flame is effective in burning a substantial portion of the soot particles that account for most of the flame radiation. Flame radiation values obtained with the high-aromatic-content Jet A and the uncoated liner were considerably higher than those obtained with the lower-aromatic-content Jet A fuel. The radiation obtained with the coated liner was approximately the same for both fuels. The measurements also showed that coated liners slightly reduced smoke concentration. For example, at conditions simulating takeoff with Jet A fuel, the SAE smoke number was reduced from 33.2 for the uncoated liner to 28.5 for the coated liner. Other combustor tests at NASA (refs. 20 and 21)

show that the TBC can reduce pollutant emissions. This is attributed to the hotter ceramic surface compared to bare metal wall reducing the quenching of CO and HC consumption reactions.

Contracted and Other Studies

In addition to the research conducted at the Lewis Research Center, studies to further understand, evaluate, and improve the TBC have been conducted under contract with industry or at no cost to the government.

Analytical studies (refs. 22 and 23) were conducted to evaluate the design, performance, and economic effects of applying the TBC to industrial gas turbines. In reference 22, it was predicted that the use of TBC could reduce power production costs by about 6 percent and give significant fuel savings. Reference 23 showed that for combined cycle powerplants, the largest payoff in efficiency increases and electric cost reductions were associated with increases in turbine inlet gas temperature permitted by the use of the TBC. The use of TBC, however, was stated to also be attractive for simple and recuperated cycles. Both of these references suggest research and technology plans for developing commercial readiness of the coating for application to gas turbine engines.

An analytical investigation was conducted (ref. 24) to determine the effects of having the TBC on the first stage turbine blades of a military aircraft gas turbine engine. Heat transfer and elastic stress analyses were performed at 10 and 50 percent span spanwise locations. Both out-of-plane and in-plane elastic stress analyses were conducted. Maximum strain ranges were calculated for the coating during a typical engine transient cycle. Results of the analyses show that the highest strain ranges in the coatings occurred at the leading edges of both spanwise locations. The magnitudes of these strain ranges when compared with that allowed for the coatings indicated that the coating would be expected to fail essentially over all areas of the airfoil. Because this had not been observed in previously-tested blades, it was recommended that coatings be examined for possible microcracking, which could provide stress relief for the ceramic while allowing it to remain attached to the blade.

Coated first stage air-cooled turbine blades were tested in a commercial aircraft engine at accelerated cyclic endurance conditions, and the results were analyzed in reference 25. Inspection of the blades after 39 hours (327 cycles) showed coating loss at the leading edge. The maximum gas temperature for these tests was 1700 K and the gas pressure 2.2 MPa. A typical cycle was 2 minutes at takeoff power and 5 minutes at idle power. The cyclic tests on these coated blades were continued for an additional 225 hours (1097 cycles).

This additional testing resulted in the loss of about one-third of the coating on the pressure (concave) side of the airfoil near the 70 percent span. Visual examination indicated that the TBC was not damaged at other locations of the airfoils or on the platforms after an accumulated total of 264 hours (1424 cycles) of engine testing. Heat transfer and mechanical and thermal stress analysis were made of the coated blade. These analyses indicated that the early coating failure occurred at the region of highest metal temperature and that the coating failed in regions where the compressive strains were the greatest, that is, at the leading edge and pressure surface of the blade.

Another contractual study is currently underway to determine the feasibility of developing an automated system to control plasma spray deposition of the TBC on gas turbine blades. This is being done to overcome the variability in deposited thickness and composition caused by manual coating application. The system which has been built integrates a multiaxis blade-handling fixture, an optical instrument for coating thickness monitoring, the plasma spray equipment, and a microprocessor-based system controller. The system is currently being evaluated. Micrographs of selected coatings deposited by the system on a turbine blade indicate uniformity of thickness and good overall structural integrity.

Substantial interest has been expressed by both the aerospace and nonaerospace industries in the use and evaluation of the TBC on their products. In many of these areas, when the potential results were assessed to be of benefit to the government, NASA has coated a small quantity of parts for which the companies supplied the results of their tests to NASA at no cost. The results of these tests on coated cooled parts from military and commercial gas turbines, military ground-vehicle gas turbines, industrial gas turbines, and Diesel engines are summarized in reference 16.

The results of this cooperative effort with industry have demonstrated the benefits of the coating in a variety of applications and the results support prior laboratory test results and heat-transfer analysis which showed that the TBC is highly adherent and that it can substantially reduce the temperature of cooled metal parts. The TBC increased part life, and eliminated the burning, melting and warping that occurred with some of the uncoated metal parts. Despite local loss of the ceramic layer during some of the tests, no melting or burning of the metal occurred in these areas. This demonstrated that local coating loss will not necessarily result in damage to metal parts.

CONCLUDING REMARKS

This summary and update on the NASA thermal barrier coating (TBC) has discussed the potential benefits of its use in a number of applications, described various research that has been conducted to better understand it, define its properties, and improve and evaluate its adherence and durability in various environments.

A number of examples have been presented of its adherence and durability in a variety of test conditions of high gas temperature and pressure at both steady state and cyclic operation. The TBC has survived several thousands of cycles and hours of operations in test rigs and over a thousand cycles and few hundred hours in engine operation. In some cases, however, local areas of the coating spalled or eroded.

Although much has been done, considerable research effort is still required to develop the coating and obtain the technology that is adequate for commercial utilization of the coating.

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Table 1. - Cyclic furnace evaluation of various zirconia thermal-barrier coatings on
Ni-16Cr-6Al-0.6Y bond coating

Alloy	Cycles to failure ^a - First visible crack, spall, etc.			
	ZrO ₂ -12Y ₂ O ₃	ZrO ₂ -3.4MgO	ZrO ₂ -5.4CaO ^b	ZrO ₂ -5.4CaO ^c
DS MAR-M-200 + Hf	^d ₆₇₃	460	255	78
MAR-M-200 + Hf	^d ₆₅₀	450	255	87
MAR-M-509	^d ₅₅₈	450	196	76
B-1900 + Hf	^d ₆₂₈	438	226	--

^aCycle, 1 hr at 1248 K and 1 hr to cool to 553 K.

^bPartially stabilized zirconia derived from ZrO₂ and CaCO₃ spray powders (cubic and monoclinic phases).

^cTotally stabilized zirconia derived from stabilized spray powder (cubic phase).

^dNo failure observed.

FIGURE LEGENDS

Figure 1. - Thermal-barrier-coated turbine blade.

Figure 2. - Comparison of calculated and measured midspan leading-edge wall temperatures of uncoated and zirconia-coated turbine vanes operating in a research engine. Inlet gas temperature, 1644 K, inlet gas pressure, 3 atm; coolant temperature, 319 K.

Figure 3. - Effect of ceramic coating on kinetic energy loss.

Figure 4. - Thermal-barrier-coated turbine blades after 500 cycles of testing.

Figure 5. - Microstructure of NASA Thermal Barrier Coating on turbine-blade leading edge at midspan after cyclic engine tests. $\times 150$

Figure 6. - Hot-gas test rig with coated air-cooled test blade.

(a) Surface temperature, 1553 K; maximum oxide layer thickness, 0.038 centimeter.

(b) Surface temperature, 1463 K; maximum oxide layer thickness, 0.028 centimeter.

Figure 7. - Cyclic performance of stabilized zirconia thermal-barrier coatings on air-cooled turbine blades in a hot-gas rig. Substrate temperature, 1188 K.

Figure 8. Yttria-stabilized zirconia thermal-barrier coating lives at high surface temperature. Oxide thickness, 0.051 centimeter.

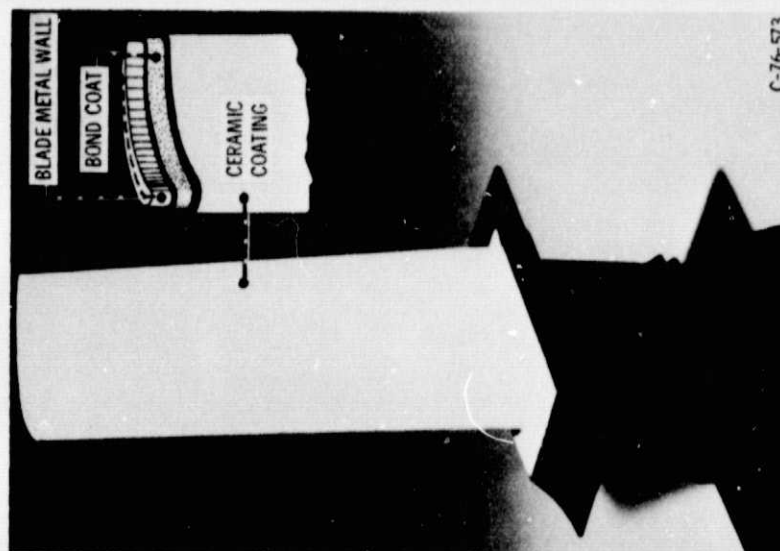
Figure 9. - Measured emittance of the NASA Thermal Barrier Coating.

Figure 10. - Total hemispherical emittance of NASA Ceramic Thermal Barrier Coating as function of temperature and thickness.

Figure 11. - Thermal-barrier-coated combustor liner.

Figure 12. - Effect of thermal-barrier coating on maximum liner temperatures; fuel, Jet A.

Figure 13. - Effect of thermal-barrier coating on flame radiation; fuel, Jet A.



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Figure 1. - Thermal-barrier-coated turbine blade.

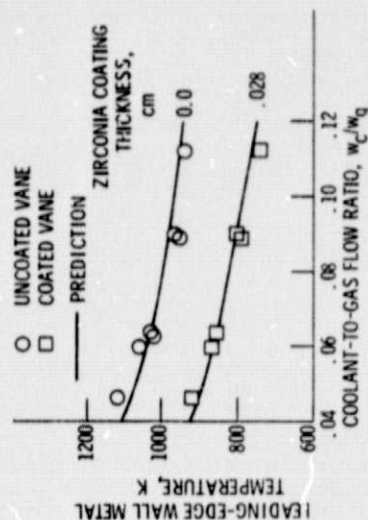


Figure 2. - Comparison of calculated and measured midspan leading-edge wall metal temperatures of uncoated and zirconia-coated turbine vanes operating in a research engine. Inlet gas temperature, 1644 K; inlet gas pressure, 3 atm; coolant temperature, 319 K.

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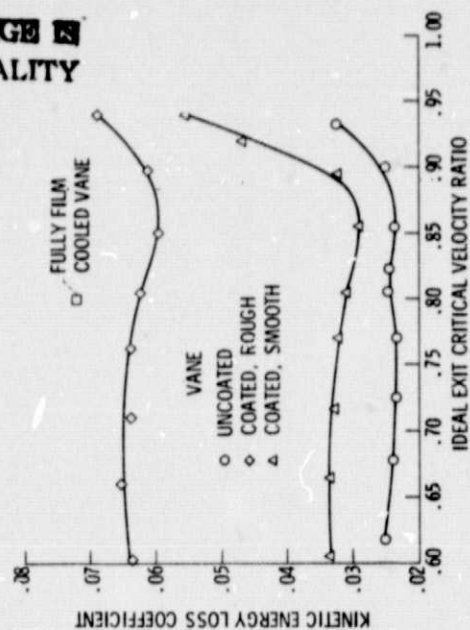


Figure 3. - Effect of ceramic coating on kinetic energy loss.

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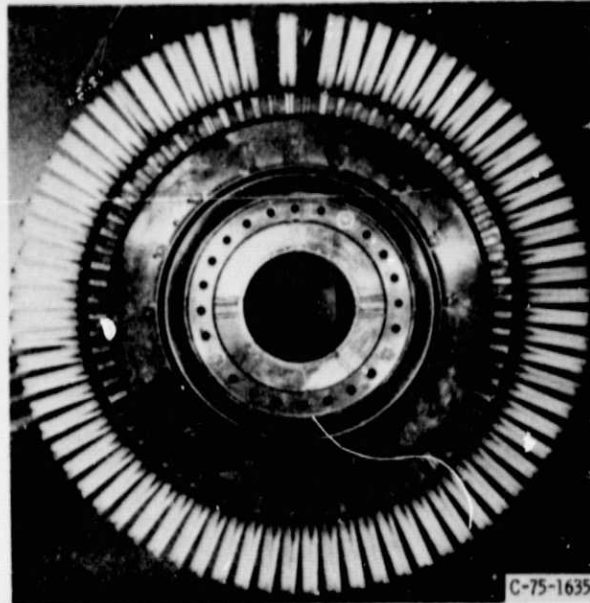


Figure 4. - Thermal-barrier-coated turbine blades after 500 cycles of testing.

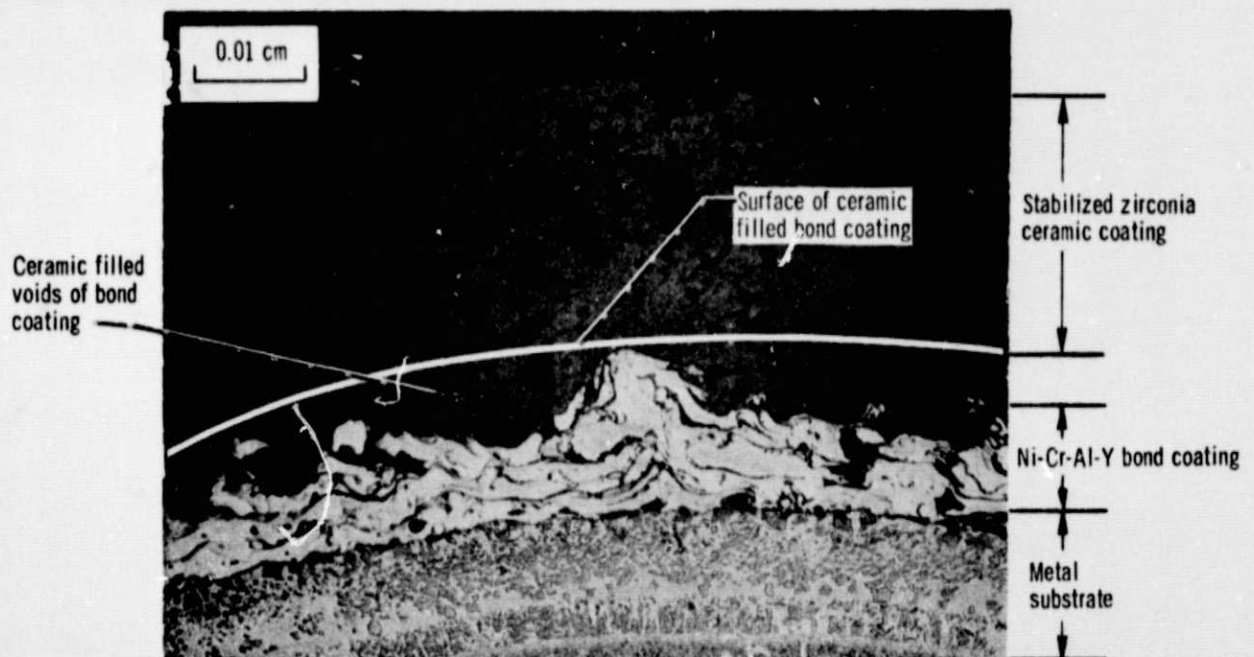


Figure 5. - Microstructure of NASA Thermal Barrier Coating on turbine-blade leading edge at midspan after cyclic engine tests. X150.

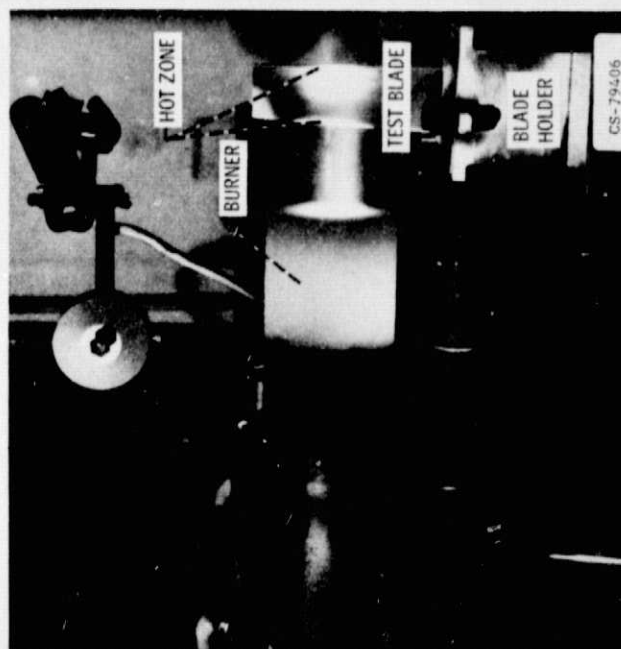


Figure 6. - Hot-gas test rig with coated air-cooled test blade.

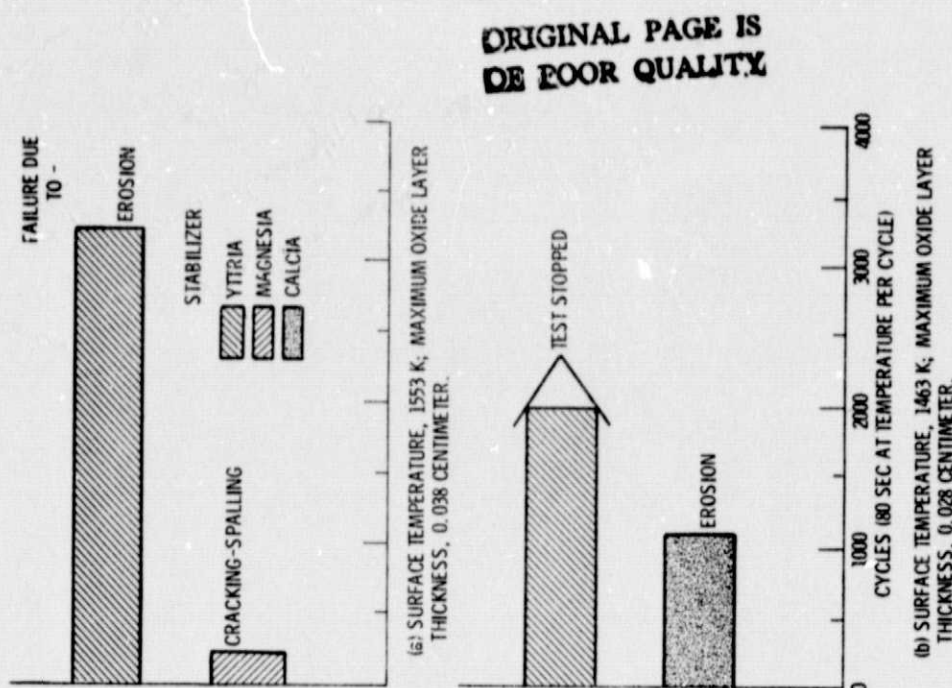


Figure 7. - Cyclic performance of stabilized zirconia thermal-barrier coatings on air-cooled turbine blades in a hot-gas rig. Substrate temperature, 1188 K.

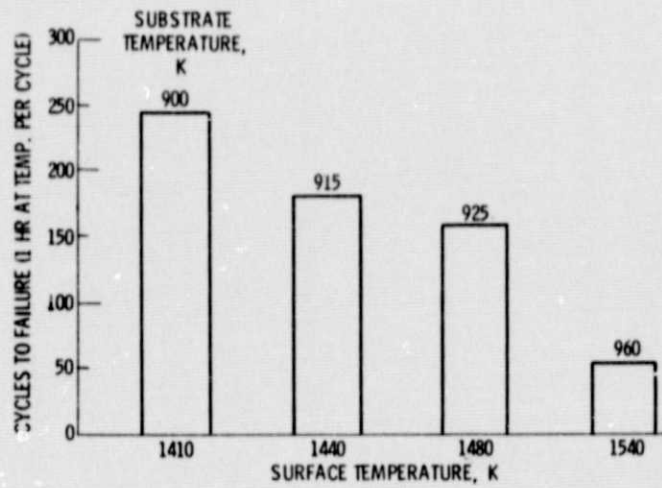


Figure 8. - Yttria-stabilized zirconia thermal-barrier coating lives at high surface temperature. Oxide thickness, 0.051 centimeter.

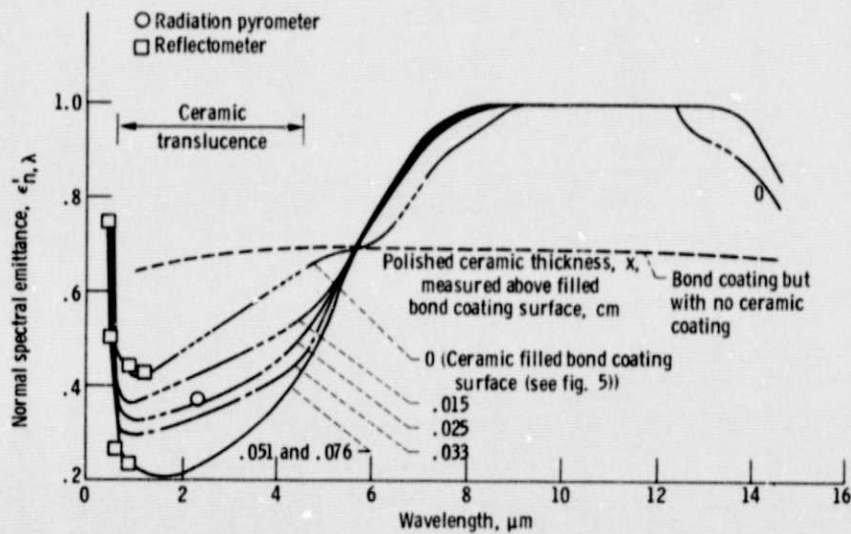


Figure 9. - Measured emittance of the NASA Thermal Barrier Coating.

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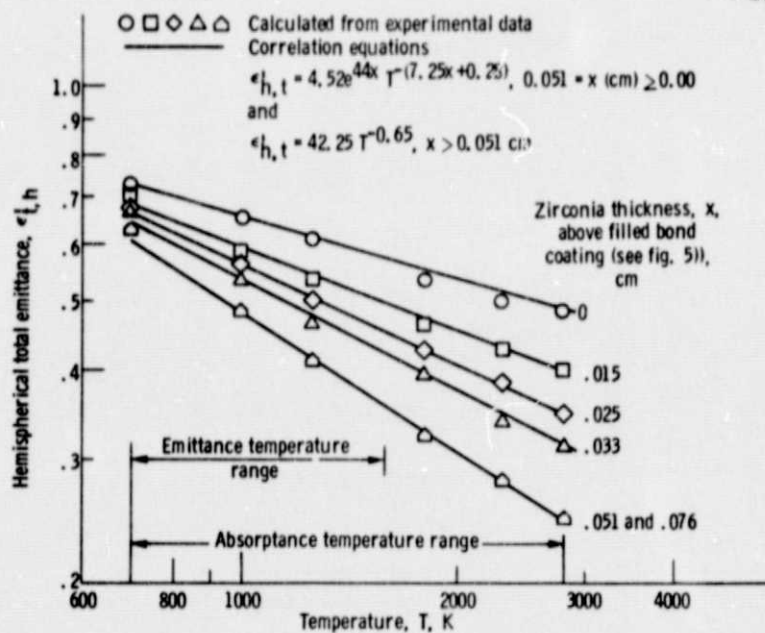


Figure 10. - Total hemispherical emittance of NASA Ceramic Thermal Barrier Coating as function of temperature and thickness.

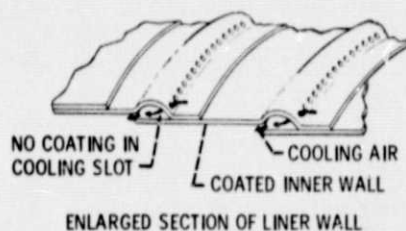


Figure 11. - Thermal-barrier-coated combustor liner.

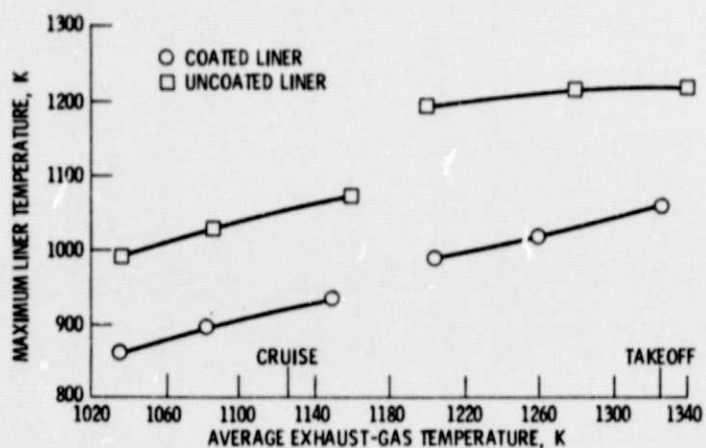


Figure 12. - Effect of thermal-barrier coating on maximum liner temperatures; fuel, Jet A.

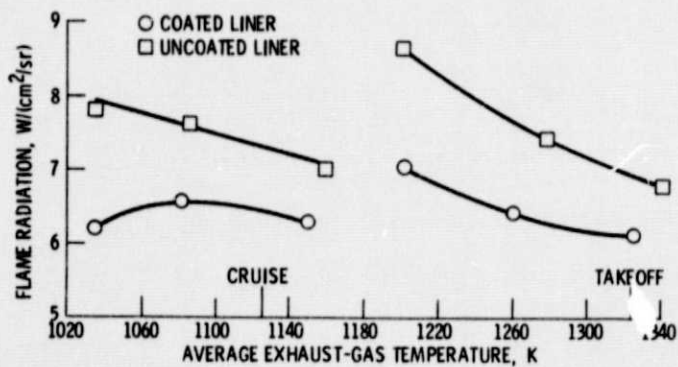


Figure 13. - Effect of thermal-barrier coating on flame radiation; fuel, Jet A.